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Planetary Mission Performance for Small Solar Electric Propulsion Spacecraft

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PLANETARY MISSION PERFORMANCE FOR SMALL SOLAR ELECTRIC PROPULSION SPACECRAFT†

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This paper presents a survey of the performance of small low-power Solar-Electric-Propulsion (SEP) spacecraft for planetary missions. These missions include main belt asteroid rendezvous, comet rendezvous, outer planet orbiters, Pluto flyby, solar probe and Mars and Mercury orbiters. Net delivery capability is presented for these missions based on current performance of both solar array and ion propulsion systems. This paper identifies those planetary missions where SEP would be most valuable, The most attractive missions appear to be small body rendezvous missions, high energy outer planet missions or a Pluto flyby mission,

INTRODUCTION

The National Aeronautics and Space Administration (NASA) is currently examining small low-cost planetary missions [1] under the Discovery program. The intent is to develop a program of small relatively inexpensive planetary missions that would complement larger planetary missions and provide more frequent mission opportunities for the science community. The Discovery program would use small chemical propulsion spacecraft with a dry mass in the range of 100-400 kg launched by a medium class launch vehicle such as a Delta II (7925).

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Although many planetary missions can be performed by small spacecraft using conventional chemical propulsion, some missions such as rendezvous missions to mainbelt asteroids or to short period comets may require long complex multiple gravity assist trajectories to deliver even modest science payloads.[2] Since one of the criteria used in selecting Discovery class missions is that of total mission time, it may be difficult to perform these small body rendezvous missions and satisfy the Discovery guidelines.

Mission studies at JPL[3,4] and elsewhere have demonstrated the potential benefits of using a spacecraft powered by Solar Electric Propulsion for many planetary missions, in particular small body rendezvous missions. Historical impediments to the use of these SEP systems for planetary missions has been both development cost and risk due to the uncertainty in the advertised performance, reliability and lifetime of these systems. Advances in electric propulsion thruster technology together with a proposed flight test of a Xenon ion thruster on an Air Force mission (SSTAR) could do much to reduce both the cost and risk of using ion propulsion systems for planetary missions.

The current interest in performing small, low-cost planetary missions has spurred the examination of using relatively low-power 5-10 kW SEP spacecraft for these planetary missions. The propulsion system for these spacecraft would likely employ either segmented ion thrusters proposed by J. Brophy at JPL[7] or 30 cm Xenon thrusters used in SEP mission studies over the past ten years by JPL and NASA Lewis Research Center (LeRC). Although probably not a Discovery class spacecraft, these low power SEP spacecraft would use a Delta 11 launch vehicle or a larger intermediate class launch vehicle such as an Atlas HAS.

There are advantages and disadvantages in using SEP powered spacecraft for these small planetary missions. A SEP system can enable a small body rendezvous mission without the use of a time consuming gravity assist trajectory mode. Mainbelt rendezvous missions, for example, can be performed in 1,5 to 2.5 years as compared with 3-5 years or more required for gravity-assist ballistic missions which deliver the same payload. Larger payloads and a more diverse set of targets are also achievable for small body missions using SEP. In addition more frequent launch opportunities are possible for many mainbelt asteroid targets since the transfer trajectory is not constrained by the ephemeris of the intermediate gravity assist body as required for a ballistic asteroid rendezvous mission.

Disadvantages to the use of SEP include concerns regarding the reliability of the thrusters that goes with the extended thrust times that are required for SEP missions and the more intensive navigation and guidance functions required during mission operations. The environmental interaction between the electric propulsion thrusters and the science payload is an additional issue that also must be considered.

SEP powered spacecraft can be used for planetary missions other than small body missions such as terrestrial or outer planet orbiter missions although the flight time and delivery capability of SEP as compared with conventional chemical propulsion is not as great for these missions as for that of small body rendezvous missions. Outer planet SEP orbiter missions still require some form of chemical propulsion for the orbit insertion maneuver since the SEP solar array can not provide the necessary power for operation of the electric propulsion system at the large heliocentric distances of these planets. For these orbiter missions the SEP system can be likened to a high energy upper stage augmenting the launch vehicle. The SEP propulsion system and possibly the solar array could then be jettisoned following final thrust cutoff in order to reduce the burden on the retro propulsion system.

Near Earth asteroid rendezvous missions can also be performed by these small SEP spacecraft although it is likely that the payload provided by a small chemical propulsion spacecraft would be adequate. A better use for these small SEP systems would be for a near Earth asteroid sample return mission however this type of mission is not covered in this paper. A small SEP system can also perform a Mercury orbiter mission in as short a transfer time as 1.5-2.5 years although the thermal environment at the distance of Mercury from the sun might present difficulties in the design of the spacecraft that could more than offset any performance advantage of SEP.

One of the major costs for planetary missions is that of the launch vehicle, hence it makes sense to compare the SEP and ballistic missions using the same launch vehicle. If similar payloads are available for a particular mission for these two propulsion options, then a further consideration would be the cost differential between the SEP and ballistic missions. Thus the SEP would need to have a distinct advantage over an equivalent ballistic mission in order to justify its additional cost. In this case one advantage could possibly be in a large reduction in mission time.

On the other hand the use of SEP may enable a particular mission, for instance a Pluto flyby mission, using a smaller and less expensive launch vehicle which would deliver the same payload to the target as a ballistic mission using a larger launch vehicle. In this case the additional cost of the SEP system and mission operations would need to be less than the additional cost of the larger launch vehicle required for the ballistic mission.

PERFORMANCE ASSUMPTIONS

The performance assessment of these planetary missions is based on the use of existing or current electric propulsion system technology. Conservative estimates of power and propulsion system mass arc used partly to account for redundant thrusters and power processors add ed to assure reliable operation of the thrust system over the life of the mission. The propulsion system mass consequently appears large when compared with technology projections of performance and mass for future SEP systems.

A simplified model of SEP system parameters is used to generate the estimates of performance presented in the following sections. This model assumes operating the thrusters at a constant specific impulse and efficiency with the variation of array output power reflected only in changing the thrust level. These effective values of specific impulse and efficiency were selected to give equivalent performance for the Vesta asteroid rendezvous mission used as an example in Reference 7. In reality these effective values vary depending upon the array power level, thruster configuration and mission. Actual SEP thrust system configurations may also vary for different missions depending upon the number of thrusters required for lifetime and redundancy.

The performance estimates to be presented for the various planetary missions thus should not be taken too literally but used primarily to provide an indication of mission feasibility. Table 1 below presents the basic vehicle and propulsion parameters which have been employed in this performance assessment. Launch vehicle performance is based on that used in current JPL mission studies.[8]

Table 1
PROPULSION AND VEHICLE PARAMETERS

Thruster specific impulse, seconds	3000
Thruster efficiency, percent	60
Power and propulsion system mass, kg	$30\ P_0 + 120$
Propellant tankage allowance, kg	$.15\ M_p$
Launch vehicle contingency, percent	10
Spacecraft adapter allowance, kg	$.05\ M_0$
Spacecraft housekeeping power allowance, kW	.25

In the above table P_0 is the solar array output power measured at 1 AU used in the trajectory simulation. Beginning-of-life array power would need to be around 10-20 percent higher in order to account for environmental and other degradation factors. M_0 and M_p arc the initial wet spacecraft mass and the consumed propellant respectively. The power and propulsion system mass reflects the addition of redundant thrusters required to meet lifetime and reliability requirements.

A requirement of a net spacecraft mass including the science payload of around 200 kg is typical for these small planetary missions. In order to accommodate a finite launch window and to allow for additional launch vehicle and SEP performance contingencies, a net spacecraft requirement of 300 kg is used as a criterion for a viable SEP mission.

The following sections present a description of each of the SEP planetary missions under consideration and describes briefly the differences between SEP and ballistic missions, The missions are arranged such that the most attractive SIN' missions (compared with ballistic) are considered first, Comprehensive comparisons of SEP and ballistic missions are not made in this paper since the entire mission scenario, including costs and mission operations, must be considered in any realistic evaluation and not just the trajectory and delivery capability as presented in this paper.

SMALL BODY RENDEZVOUS MISSIONS

The most attractive use of SEP would appear to be for rendezvous missions to either mainbelt asteroids or to comets where the post launch energy requirements for a direct ballistic mission are large. Direct ballistic trajectories to these small, nearly massless bodies can be expensive in terms of launch energy and post launch AV. Consequently it has been necessary to utilize gravity assists of one or more intermediate bodies to deliver the required payload to these small body targets. The use of these intermediate gravity assist bodies results in a reduction in launch energy which allows use of a smaller launch vehicle. A reduction in the post-launch AV can also result in particular for mainbelt asteroid missions. Although this gravity assist trajectory technique enables ballistic missions to many of these small bodies, the resulting mission times are long as compared with that of a direct mission. The use of these intermediate gravity-assist bodies also has the effect of constraining the launch opportunities available for a particular target.

It may not be necessary for the thrusters to utilize all the available power from the solar array heliocentric distances of around 1 AU for these small body rendezvous missions, Although the total delivered spacecraft mass decreases slightly when the maximum array power is not utilized, the SEP system can operate with a fewer thrusters. Thus there could be a net increase in useful payload since the SEP system could be designed with fewer thrusters and power processors. The actual decrease in system mass will depend upon reliability, lifetime and redundancy in design of the thrust subsystem. These issues of SEP system design are dependent upon the particular thrusters being, used and are not addressed in this paper, the only concession being to constrain the maximum thruster input power to the solar array power level at 1 AU for missions which thrust inside the orbit of the Earth.

MAINBELT ASTEROID RENDEZVOUS MISSIONS

A comprehensive examination of the delivery options available for ballistic mainbelt asteroid rendezvous missions has been made at JPL by Chen-wan Yen.[9] This examination indicated a launch energy requirement ranging from a $C_3 = 25$ to 40 km^2/s^2 and a post-launch AV requirement of 4 to 5 km/s for direct ballistic rendezvous missions to the inner mainbelt asteroids. The result of using a SEP system for these direct missions is two-fold; first the launch energy is reduced by a factor of 2-4 times and second, the AV that the SEP must contribute is at least double that of the ballistic mission, The reduction in launch energy for the SEP mission compared with the corresponding direct ballistic mission occurs because the higher specific impulse of the SEP enables the SEP system to contribute more efficiently to the total required mission energy. Although the AV that the SEP must contribute to the mission is much greater than that for the corresponding direct ballistic mission, this AV is achieved much more efficiently since the specific impulse of the SEP thrusters is at least a factor of 10 greater than the specific impulse for a chemically propelled spacecraft.

An important consideration for these inner mainbelt asteroid rendezvous missions is the array power available at the terminal portion of the mission for rendezvous. Thrusters typically have a minimum power requirement below which they cannot operate reliably, hence for a rendezvous mission to an asteroid that lies near the outer part of the inner main belt, such as the largest asteroid Ceres, it is necessary to size the solar array to a power level sufficient to provide power to the thrusters at the farthest expected heliocentric operating distance.

An array power level of 5 kW is sufficient to deliver the desired net spacecraft mass of 300 kg to many inner mainbelt asteroids with a semi-major axis 2.5 AU or less. Some examples of missions to large inner mainbelt asteroids including several to one of the more scientifically interesting targets, 4-Vests, are presented in Figure 1 for launch opportunities ranging from 1999 through 2008. These 5 kW SEP missions use a Delta II (7925) launch vehicle and have a net rendezvous spacecraft mass of 300 kg, Approximately the same net mass capability is available for ballistic chemical propulsion gravity assist missions to these same bodies but at much longer flight times..

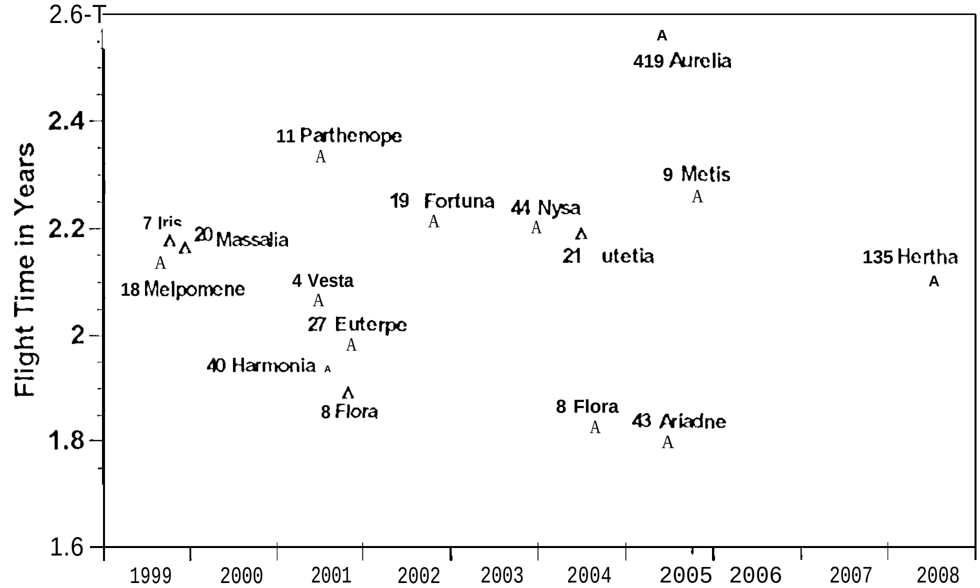

Figure 1 Mainbelt Asteroid Rendezvous 5 kW SEP Delta II (7925)

A Delta 11 launch vehicle and a 5 kW SEP system are not capable of delivering this same 300 kg spacecraft to asteroids much farther out than around 2.5 AU in the inner main belt such as to the largest asteroid, 1 -Ceres. By increasing the solar array power to 8 kW, a maximum delivery capability of 200-250 kg to some of the more difficult targets such as 1-Ceres at flight times of up to around 3 years is possible. This maximum performance is primarily constrained by the injection capability of the launch vehicle and an increase of array power 10 kW and a move to a larger launch vehicle such as an Atlas IIAS enables rendezvous missions to these more distant asteroids. Several examples of rendezvous missions to these more distant asteroids are presented in Figure **2.**

Note that the flight time is generally shorter for the missions shown in Figure **2** than for the less energetic missions presented in Figure 1. 'I'he missions presented in Figure. 1 are actually pushing the performance of the 5 kW SEP and Delta II launch vehicle and require a longer flight time to deliver the required 300 kg spacecraft mass. Increasing the array power level to 8 kW and using the same launch vehicle allows some decrease in flight time. On the other hand the missions shown in Figure 2 using an Atlas IIAS launch vehicle are relatively less demanding on the SEP system and result in a shorter flight time. Although these missions would be more costly because of the larger launch vehicle and higher powered SEP, they do indicate that aggressive mainbelt asteroid missions are possible with these small SEP systems.

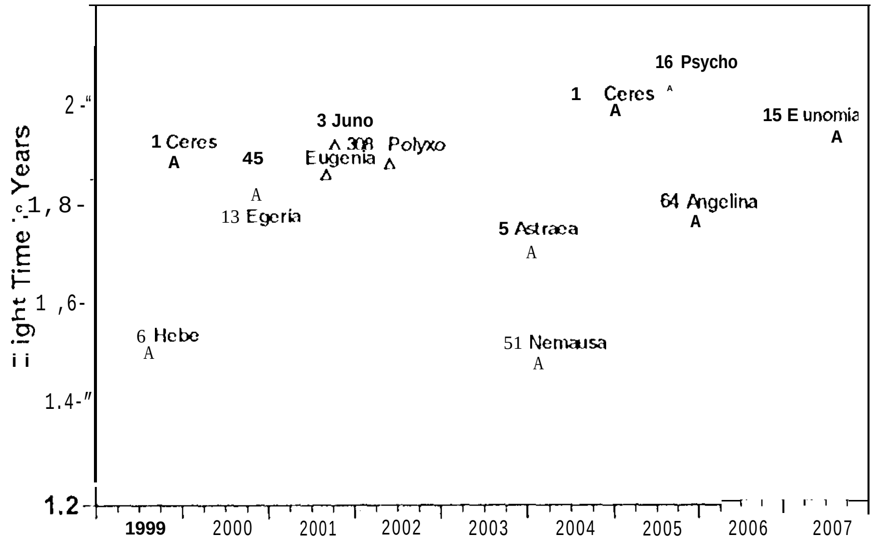

Figure 2 Mainbelt Asteroid Rendezvous 10 kW SEP Atlas IIAS

The missions presented in these first two figures represent only a small sample of possible asteroid rendezvous missions available during the considered time span. There is sufficient performance margin available for many of these asteroid rendezvous missions to include additional rendezvous with other asteroids. Multiple asteroid rendezvous trajectories have not been addressed in this paper but considerable work has been done in the past to illustrate this SEP capability.[10]

The use of a SEP power spacecraft can also increase the number of launch opportunities available for some of the more interesting asteroid targets. If these bodies traveled in circular orbits, the performance would be about the same for each opportunity which occurs once every synodic period of the asteroid and Earth. However asteroids are generally in moderately eccentric orbits and the variation of performance from launch opportunity to launch opportunity can vary significantly. Rendezvous missions to 4-Vests have been studied in greater detail than to any of the other asteroids because of the greater scientific interest in this asteroid. In order to illustrate the variation of performance for different launch opportunities, missions to 4 -Vesta were examined for launch opportunities covering an eight year span starting in the year 2000. Some variation in performance is to be expected since the orbit of 4 -Vesta is inclined around 7 degrees to the ecliptic and has an eccentricity of around .09.

The results of this examination are presented in Figure 3 where the flight time for each of the launch opportunities is shown for a delivered net

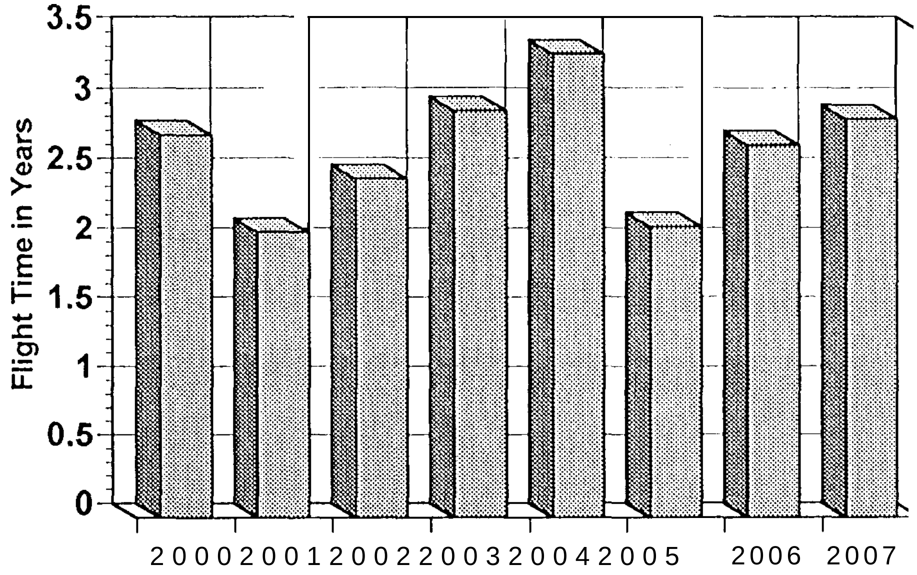

Figure 3 Rendezvous opportunities for 4-Vests

spacecraft mass of 300 kg using a 5 kW SEP system launched on a Delta II launch vehicle. Both the 2001 and 2005 opportunities are attractive and have flight times slightly over 2 years. Both these opportunities have a second launch opportunity, not shown, with a flight time in the range of 2.6-2.7 years. The variation in performance between different launch opportunities is primarily a function of the orbital eccentricity of the target asteroid. Asteroids in more eccentric orbits than that of 4-Vests should show a greater variation between the launch opportunities.

COMET RENDEZVOUS MISSIONS

Although direct ballistic rendezvous missions to comets and mainbelt asteroids have total mission energies that are not significantly different, the division of mission energy between the launch and post-launch mission phases is different for these two types of missions because of the much higher orbital eccentricity of short period comets. 'l'he best performance for either ballistic and SEP comet rendezvous missions is to comets with orbits having low inclination and a perihelion distance around 1 AU. Most of the mission energy for ballistic missions can be obtained from the launch phase of the mission and a correspondingly smaller part during the rendezvous phase. Ballistic gravity assist trajectories for these comet rendezvous trajectories essentially extend the launch phase to include the gravity assist, maneuvers of either or both the Earth and Venus. Consequently the launch energy is relatively low for these ballistic gravity assist trajectories and the additional gravity assists in effect add additional energy to the "launch" phase. in order to keep the post launch

AV as low as possible, these ballistic missions often have rendezvous maneuvers near comet aphelion at a distance of 5 AU or greater.

SEP comet rendezvous missions have thrusting constraints similar to those of mainbelt asteroid rendezvous missions in regard to thrusting at extended heliocentric distances. If thrusting is constrained to distances of 2.5 to 3 AU or less, comet rendezvous must occur within 1 year of comet perihelion. In order to provide sufficient performance for these comet rendezvous missions, thrusting near comet perihelion is necessary and rend ezvous with the comet generally must occur after comet perihelion. Although science requirements generally dictate a pre-perihelion rendezvous as early as possible, the degradation in performance or flight time for a SEP pre-perihelion rendezvous makes such a mission difficult.

An example of a SEP comet rendezvous trajectory to the short period comet Kopff is shown in Figure 4. The trajectory mode shown in this figure is called an indirect, post-perihelion rendezvous, the term indirect indicating that the SEP makes more than one full revolution about the sun during the initial phase of the mission. This trajectory mode offers a great deal of

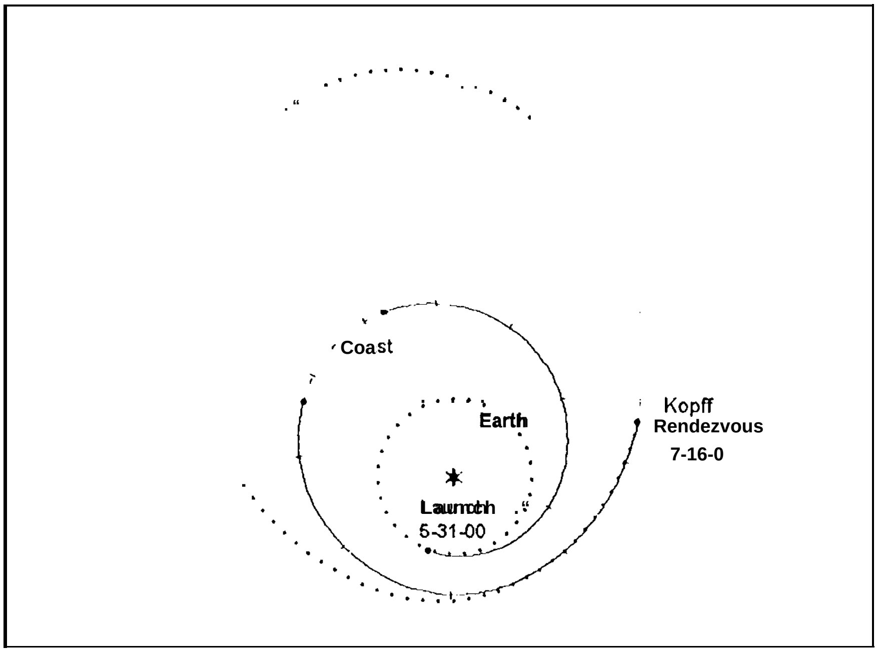

Figure 4 Kopff Post-Perihelion Indirect Rendezvous

flexibility for comet rendezvous missions since the phasing between the Earth and the comet is compensated for by varying the aphelion distance on the first phase of the mission following launch.

Expected net spacecraft delivery capability for a number of short period comets is shown in Figure 5. The trajectories shown in this figure were calculated for a 5 kW SEP and a Delta II launch vehicle. The location of the comet rendezvous on the orbit was either optimized or occurred at a thrust cutoff distance of 2.5 AU where the available array power dropped to a point insufficient to operate the thrust system. There are a number of comet rendezvous launch opportunities shown, including some CRAF targets, which have acceptable performance during the seven years covered by this figure.

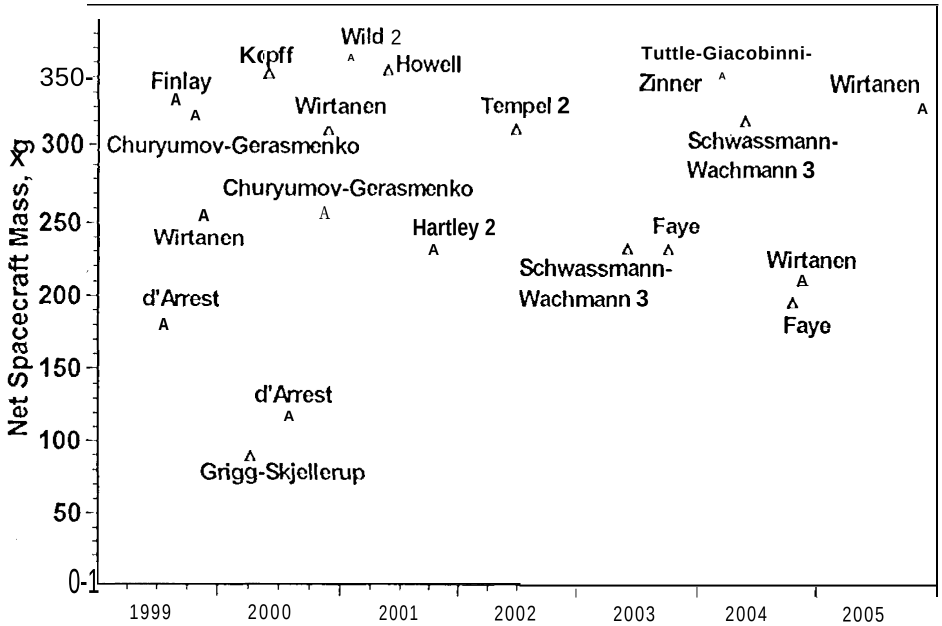

Figure 5 Comet Rendezvous Launch Opportunities

Another potential comet mission is that of a sample return mission. These have been considered in the past for higher power SEP systems and larger launch vehicles than a Delta II[11]. These sample return missions would employ the SEP system for rendezvous with the comet, Following rendezvous the SEP would be jettisoned and return from the comet to a direct atmosphere c entry at the Earth would be accomplished with a small chemical propulsion system. Delivered mass requirements at comet rendezvous for a sample return mission would likely require a 10 kW SEP and Atlas IIAS launch vehicle.

OUTER PLANET FLYBY AND ORBITER MISSIONS

As noted previously it is not practical to operate the SEP thrusters much beyond 2.5 to 3 AU from the sun because reduced solar array power may not result in reliable thruster operation. Although the solar-array power can be increased to allow thruster operation at distances beyond the above it is still not practical to use the SEP thrusters at the distances of any of the outer planets. It is also not practical to use current SEP systems on a direct transfer trajectory to the outer planets because the required launch energy is high and the resulting short thrusting time for the SEP system does not allow enough performance to gained so as to counteract the mass of the SEP system itself. These direct transfer trajectories would appear more attractive for advanced SEP concepts however.

However SEP outer planet missions can be performed using either an indirect transfer mode with a transfer angle of around 500-600 degrees or a two-year Earth gravity assist trajectory. This mode is known by the acronym SEEGA for Solar Electric Earth Gravity Assist and is the low-thrust analogue of the ballistic ΔV-EGA mode. The SEP thrusters using this mode would still shut down at a heliocentric distance well short of that of the target. At this point the SEP system could be jettisoned in order to reduce the load on the chemical propulsion system that would be required for orbit insertion, Either a small monopropellant system for flyby trajectories or a larger hi-propellant system for orbiter trajectories woul d be required.

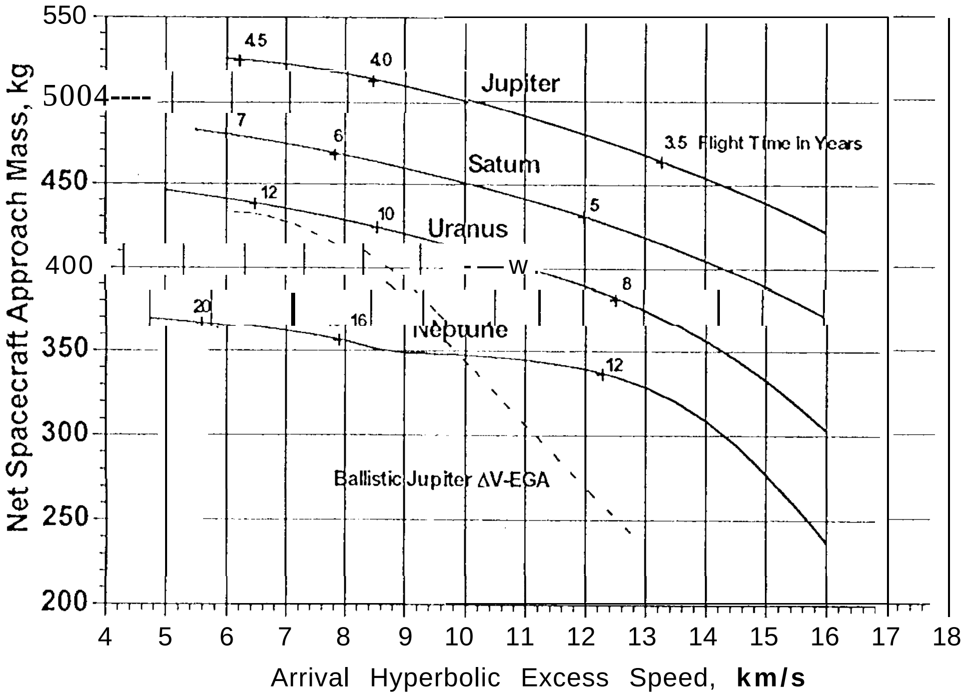

Figure 6 SEEGA Outer Planet Delivery Capability

The performance advantage of SEP as compared with ballistic is not as great if it exists at all for these outer planet missions. The performance advantage of SEP for both flyby and orbiter missions should be greater as the heliocentric distance of the target bodies increases or for higher energy transfers, This advantage occurs because the SEP is capable of adding a significant amount of energy to the trajectory by thrusting following Earth swingby. This advantage has been confirmed for a Pluto flyby mission using a small SEP launched off a Delta II vehicle. Figure 6 presents the net delivery capability to each of the four major outer planets for a 5 kW SEP and a Delta 11 launch vehicle using an Earth gravity assist mode. Net approach mass is presented rather than net mass in orbit in order to avoid specifying either the chemical retro system or the target orbit parameters. Performance for a ballistic two-year ΔV-EGA ballistic mission to Jupiter is also shown in this figure in order to illustrate the relative performance advantage of SEP.

Although performance for a Jupiter mission may be only marginally better than that of a ballistic mission at the lower arrival energies required for an orbiter mission, the SEP spacecraft shows a distinct performance advantage at the higher arrival energies at Jupiter. An example of such a higher energy mission is that of a solar probe mission that uses a gravity assist by Jupiter to put the spacecraft into an orbit inclined 90 degrees to the solar equator and passing at a distance of 4 solar radii from the sun. The hyperbolic excess speed required at Jupiter for a solar probe mission is in the range of 12.5 to 13 km/s and requires a swingby distance at Jupiter in the range of 6 to 10

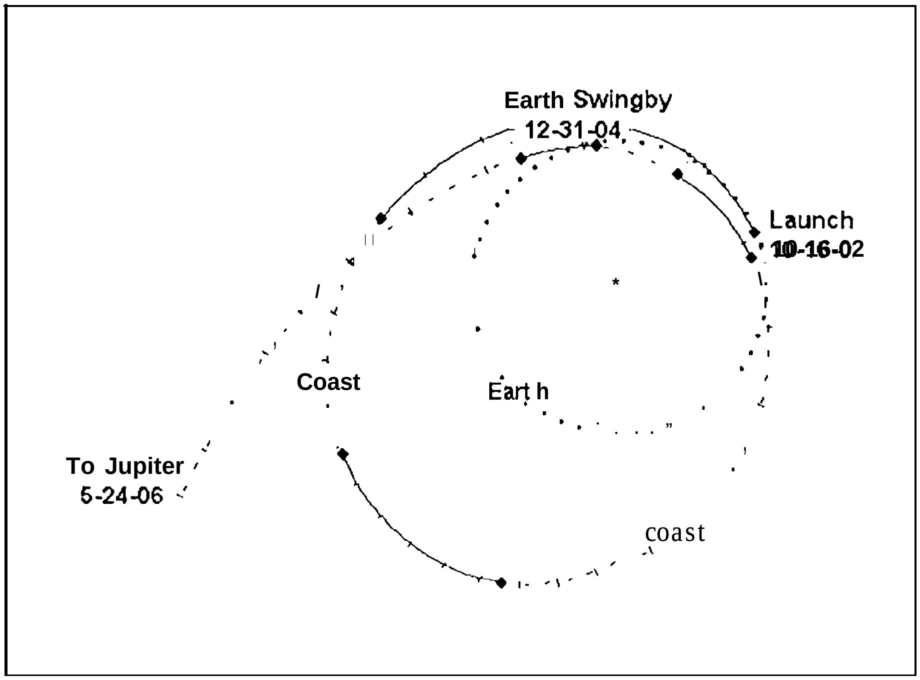

Figure 7 Solar Probe using Earth-Jupiter Gravity Assist

Jupiter radii. The net spacecraft delivery capability indicated in Figure 6 is greater than 400 kg and can be compared to a delivery capability of around 250 kg for a similar ballistic mission launched on a Delta II. The near-Earth phase of this Earth-Jupiter gravity assist solar probe mission is shown in Figure 7. This mission uses a post-perihelion Earth gravity assist and is a relatively moderate mission for this small SEP system. The solar probe mission actually requires less energy from the SEP system than does the asteroid rendezvous missions discussed previously.

PLUTO FLYBY

Another high energy mission that appears attractive for a low power SEP system is that of a Pluto Flyby. The current concept for a ballistic Pluto flyby mission requires a Titan IV/Centaur launch vehicle plus several additional solid rocket stages on the Centaur stage in order to deliver a moderate size payload on a direct trajectory to Pluto. A SEP system can deliver the same payload as this ballistic mission in a comparable flight time but required a much smaller launch vehicle than the ballistic mission. The SEP mission does, however, require a gravity assist maneuver at the Earth in order to achieve the necessary mission energy.

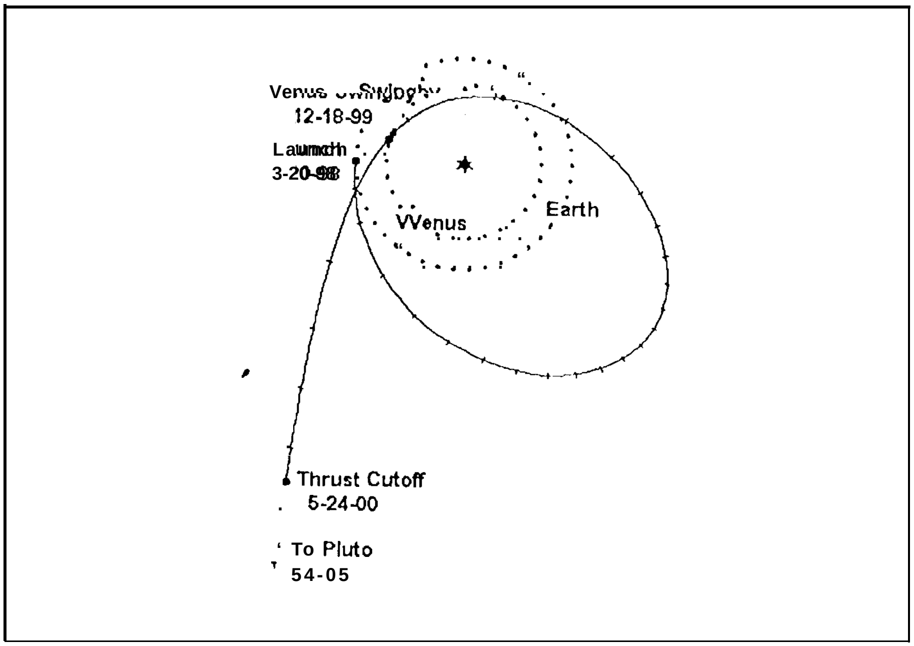

Figure 8 Pluto Flyby using Venus Gravity Assist

This mission, and likewise any other outer planet missions, would likely require RTGs for power following jettison of the SEP system. This requirement for RTGs makes the use of Earth gravity assist trajectories undesirable and alternate trajectory modes that do not involve an Earth gravity assist would be highly desirable. An alternate to using the Earth for a gravity assist is to use Venus. Using a gravity assist of Venus will constrain the launch opportunities available for outer planet missions and the Pluto flyby mission more severely than would an Earth gravity assist trajectory because the ephemeris of three bodies are now involved rather than two. An acceptable SEP Pluto flyby mission using a Venus gravity assist is shown in Figure 8.

This trajectory is quite similar to an Earth gravity assist SEP trajectory except that the spacecraft now has a perihelion inside the orbit of Venus. This fast Pluto flyby mission has a high enough mission energy that there is continuous thrusting until final thrust cutoff, This mission would likely require a 10 kW SEP system and an Atlas IIAS launch vehicle in order for the flight time to be competitive with the ballistic mission, A comparison of the performance using the above SEP and launch vehicle combination is shown in figure 9 for both Earth gravity assist trajectories and Venus gravity assist trajectories. Both pre-perihelion and post-perihelion Venus gravity assist trajectories are shown. The performance using a Venus gravity assist for this particular opportunity appears competitive with that using an Earth gravity assist.

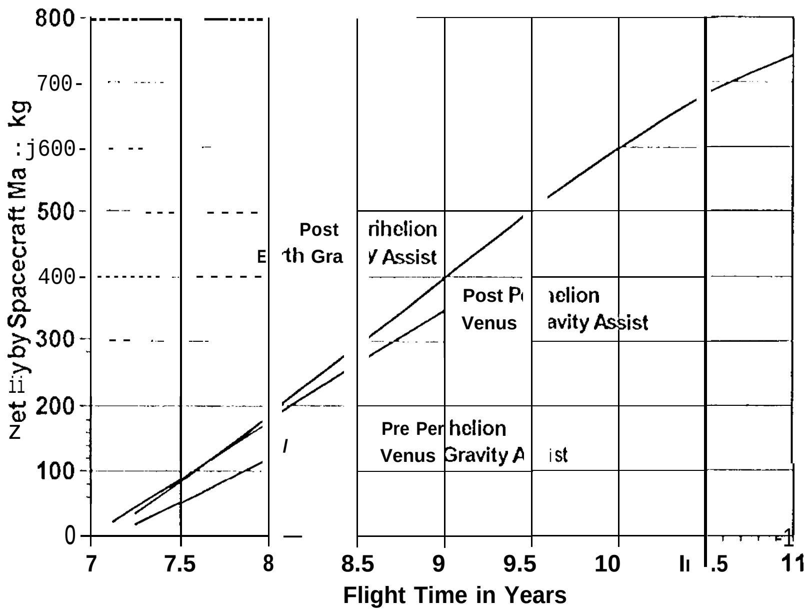

Figure 9 Gravity Assist Pluto Flyby Performance

MERCURY ORBITER MISSIONS

Mercury orbiter missions using a SEP spacecraft have been examined in the past. The 5 kW SEP system and Delta II launch vehicle are capable of delivering a net spacecraft mass of around 300 kg on a low energy approach path to Mercury. For capture into a low circular orbit about the planet, there are two options that can be considered. The first option uses the SEP system and a low-thrust spiral capture for orbit insertion. This option can deliver a net spacecraft mass of around 200 kg or more into orbit however the therms] design of the SEP and spacecraft may prove difficult for this option. A second option would be to jettison the SEP system and use a chemical retro propulsion system for orbit insertion. This second option could only deliver around 100 kg or less into a low circular orbit at Mercury.

SEP Mercury orbiter trajectories have a transfer time in the 600-750 day range and would involve around three revolutions around the Sun before encountering Mercury. These orbiter missions would require nearly continuous thrusting at maximum thruster power and result in a large propellant loading. More than double the useful net spacecraft mass would be possible if a 10 kW SEP system and an Atlas IIAS launch vehicle vere used.

VENUS AND MARS ORBITER MISSIONS

Both Venus and Mars orbiter missions are nherently ow energy missions and not as well suited for SEP since the increase in total mass delivery may not compensate for the added mass of the SEP system itself. If a requirement exists for high power for science for these orbiter missions, the overhead for a SEP system would only be that of the thrust system since the solar array would be considered part of the payload.

As for the Mercury orbiter mission, there are two options available for orbit insertion into a low circular orbit, a low-thrust spiral capture or a chemical retro maneuver. If a low-thrust capture spiral is used, the total time

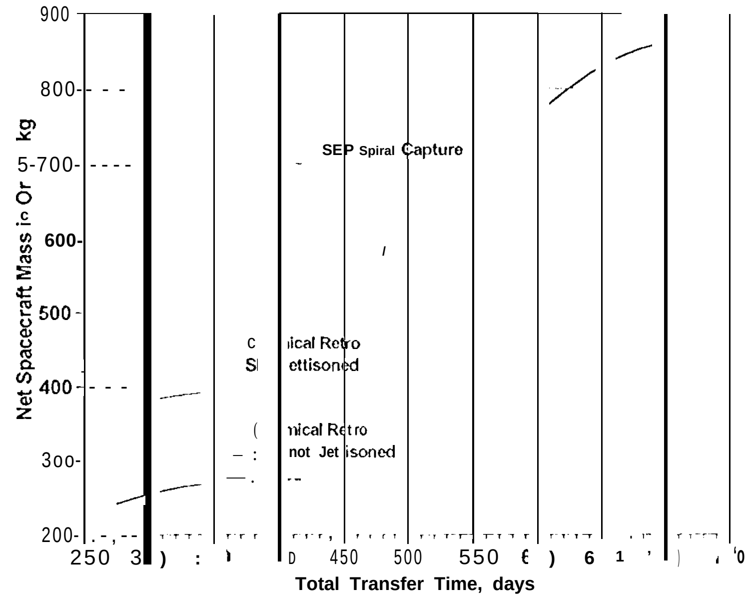

Figure 10 SEP Mars Orbiter Options

from launch to final orbit insertion is around 50% greater than if a chemical retro system were used although the net delivery capability can be nearly twice as high for the spiral capture mode. A comparison of these modes is shown in Figure 10 for a Mars orbiter mission. Two options are shown for the chemical retro capture option, one involving jettison of the SEP system prior to orbit insertion, the second retaining the SEP system through the orbit insertion.

The increase in flight time and net delivery capability is readily apparent in Figure 10 for the spiral capture option, As shown in this figure it is obviously more effective to jettison the complete SEP system prior to orbit insertion, However retaining only the solar array and jettisoning the thrust system is another option that coul d be considered and would result in a net spacecraft capability somewhere between the two lower curves.

SUMMARY

The preceding discussion mainly validates what is commonly known, that SEP has the greatest comparative advantage over conventional chemical propulsion for small body missions. Designing a common SEP system for all the above missions except perhaps the Mercury orbiter mission appears possible, the only change being in the size of the solar array for some asteroid rendezvous missions beyond the inner main belt. Although the intent of the paper is not to promote the development of a common SEP system for planetary missions, the results of the feasibility study do indicate that a variety of attractive SEP powered planetary missions are possible with a similar sized SEP systems.

Re-iterating, the mission performance presented in this paper for the small, low-power SEP systems is based upon current estimates of performance of ion bombardment thrusters. The net mass performance estimates should be used with discretion since the calculated performance is only as accurate as the models of thruster and array parameters that are used in the trajectory simulations. More reliable performance estimates for these planetary missions will depend upon better knowledge and modeling of both thrusters and solar arrays.

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